Status Report on the Penn State RBCC Test Program

J. Cramer, M. Lehman, S. Pal, S.-Y. Lee, R. Santoro
Department of Mechanical and Nuclear Engineering
The Pennsylvania State University
University Park, PA 16802

ABSTRACT

The status of the RBCC ejector mode research program at Penn State is reviewed. Recent hardware modifications and measurement system improvements are discussed, including the motivation for these changes. Results from a series of tests with a single thruster configuration at a chamber pressure of 200 psia and with an area ratio 3.3 nozzle are presented. These results indicate that the primary (rocket exhaust) and secondary (entrained air) flow streams mix much more rapidly than a previous test series with an area ratio of 6.0 nozzle. Finally, the plans for a test series with a twin thruster configuration are discussed.

BACKGROUND

The Penn State Propulsion Engineering Research Center (PERC) has been conducting a Rocket-Based Combined Cycle (RBCC) research program since 1996. This research effort is funded by NASA, and it directly supports the NASA 3rd Generation Launch Vehicle Technology Program. The 3rd Generation Technology Program has the stated goal of reducing payload launch costs and increasing vehicle safety and reliability by two orders of magnitude by 2025. An RBCC engine is considered one of the leading propulsion system candidates for achieving these goals in a 3rd generation launch vehicle.

The focus of the Penn State RBCC program is the experimental investigation of the ejector mode performance of an RBCC flowpath. An RBCC engine operates in ejector mode during the initial stage of flight, from take-off to the transition to ramjet mode. This operating regime typically covers flight speeds from Mach 0 to 3, and dynamic pressures from 0 to 1500 psf. In this mode the rocket thrusters, which are integrated into the RBCC ramjet/scramjet air duct, fire to provide the primary propulsive force. In addition, the thrusters act as ejectors, entraining air in the duct and raising the static pressure of that air. By mixing and combusting this air stream with excess fuel from the thruster (simultaneous mixing and combustion [SMC]) or from downstream fueling ports (diffusion and afterburning [DAB]), the net effect is an augmentation to the thrust generated by the rocket thrusters alone. This thrust augmentation and the corresponding increase in specific impulse are two of the features that make an RBCC system attractive for future launch systems.

The primary objective of the current research program is to develop a detailed understanding of the physical processes involved in the mixing and combustion of the primary (rocket exhaust) and secondary (entrained air) flow streams. This objective is being achieved through a combination of conventional propulsion measurement methods and advanced combustion diagnostics. A second objective of the program is to produce a comprehensive data set that can be used by analysts to develop and improve computational fluid dynamic (CFD) models for RBCC systems.

EXPERIMENTAL SET-UP

RBCC Test Hardware

The test article is a two-dimensional, heat sink copper flowpath based on the geometry of a well-documented Marquardt ejector-ramjet configuration from the mid-1960s [1]. The basic hardware configuration has been described in detail in recent reports on this program [2,3], so only a brief description is provided here. Two options are available for introducing air into the flowpath. A converging inlet section open to the atmosphere can be used to simulate sea-level static conditions (Mach 0), or an airbox can be directly connected to the flowpath to force in air at simulated flight conditions (Mach 1.0 or 1.9).

The flowpath, as depicted in Figure 1, is 109 inches long from the airbox attachment flange to the duct throat. The duct is 3 inches wide at all axial locations. The duct is 5 inches tall from the air inlet through the constant area combustor section. The duct height gradually increases from 5 inches to 10 inches in the diffuser section, and stays constant at 10 inches in the afterburner section. A converging nozzle forms a 5-inch high throat at the duct exit plane. A series of optical access windows and measurement ports are located on the top wall and the sidewalls along the entire length of the duct.

A 1.75-inch tall rocket thruster is positioned on the centerline of the duct, which allows air to flow symmetrically above and below the thruster. The thruster, which uses gaseous hydrogen (GH_2) and gaseous oxygen (GO_2) propellants, can be operated at chamber pressures up to 500 psia. It has a rectangular nozzle to provide uniform flow across the 3-inch width of the duct. The thruster nozzle throat is 0.1 inches tall and the baseline nozzle exit height is 0.6 inches. With an area ratio (AR) of 6.0, this nozzle expands the rocket exhaust from a chamber pressure of 500 psia to approximately atmospheric pressure. Gaseous hydrogen can be injected into the afterburner normal to the main flow using a series of secondary fuel ports in the sidewalls.

A series of changes to this basic hardware configuration have been implemented recently. The baseline rocket nozzle was replaced with one designed specifically for 200 psia chamber pressure operation. The nozzle exit height was reduced from 0.6 inches to 0.33 inches in order to eliminate overexpansion of the rocket exhaust. The 2-inch by 3-inch windows on the sidewalls were replaced with 2-inch by 5-inch windows. This change provides optical access from the top to the bottom of the duct at all window locations in the constant area combustor. Finally, the converging nozzle of the duct was extended 2.81 inches in the axial direction while maintaining the same convergence angle (27.8 degrees). This nozzle extension reduced the duct throat height from 5 inches to 2 inches, resulting in significantly higher pressures in the duct. Figure 2 illustrates the difference in the duct static pressure profile for these two throat sizes. This change in throat height provides a more realistic duct condition when the thruster is operated at a chamber pressure of 200 psia. One other planned change, the replacement of the single thruster with a twin thruster configuration, will be discussed in more detail later in this paper.

Measurement Systems

The test hardware accommodates conventional propulsion measurement systems. The thrust of the RBCC flowpath is measured using a load cell, and propellant flowrates are measured using critical flow venturis. For the direct connect configuration, the air flowrate is also measured with a critical flow venturi. For the sea-level static configuration, where the entrained airflow cannot be controlled, static pressure measurements in the inlet section are used to measure the air flowrate. Local heat flux measurements can be made at 25 axial locations on the duct side and top walls. In addition, there are 32 ports on the duct side and top walls for static pressure measurements.

OPTICAL DIAGNOSTICS SYSTEM

Raman Spectroscopy

After being excited by a photon at a given frequency, a molecule may release a photon at a different, but predictable frequency. This inelastic energy exchange is known as spontaneous Raman scattering [4]. Because different types of molecules release photons at distinct frequencies, Raman scattering can be used to measure the number density of various chemical species typically found in a combustion environment. For the past several years, researchers at PERC have used Raman spectroscopic techniques to measure profiles of major species concentration and gas temperature in experimental combustion devices [2,3,5,6]. The technique briefly described here is the same one used by Lehman [2,3] with a few changes that will be noted.

A frequency-doubled Nd:YAG laser (532 nm) is used to produce Raman scattering in the RBCC flowfield. This laser operates at 10 pulses per second, with pulse energies on the order of 600 mJ per pulse. The pulse width is stretched from approximately 10 nanoseconds (nsec) to 30 nsec to prevent damage to the optical access windows. The laser beam is focussed with a 1.0 meter focal length lens before it passes through an access window at the top of the RBCC duct. The resulting probe volume is a vertical line on the centerline of the duct approximately 0.040 inches in diameter. The laser beam can be routed to any of the access windows in the duct, thus providing the capability to make Raman measurements at several discrete axial positions.

The signal collection optics are mounted horizontally, normal to the flow axis. An intensified charge coupled device (ICCD) camera with a two-dimensional pixel array is used to image the laser line through an access window on the sidewall. As currently configured, the pixel array provides a spatial resolution of 0.027 inches in both directions. Different narrow bandpass filters are mounted in front of the camera lens to selectively measure each of the major combustion gas species- hydrogen, oxygen, nitrogen and water.

Data Reduction Technique

The fundamental equation for relating measured Raman signal to species mole fraction is given by [4]:

$$S_i = E_l \cdot n_i \cdot K_i \cdot f_i(T) \qquad \text{(Eqn. 1)}$$

where S_i is the measured Raman signal strength of species "i", E_l is the incident laser power, n_i is the species number density, K_i is a constant that depends on the collection efficiency of the system and the Raman cross section of the species, and f_i (T) is the bandwidth factor, which is a function of temperature. Rearranging Equation 1 and applying the ideal gas equation of state leads to an expression for the mole fraction of each species (X_i):

$$X_i = \frac{P_i}{P_t} = \frac{S_i}{E_i \cdot K_i \cdot f_i(T)} \cdot \frac{R_u \cdot T}{P_t} \qquad \text{(Eqn. 2)}$$

where P_i is the partial pressure of species "i", P_t is the pressure of the system, R_u is the universal gas constant, and T is the temperature. The assumption that the only species with significant mole fractions are H_2, O_2, N_2 and H_2O leads to the equation required for closure of the calculation:

$$\sum_{i=1}^{4} X_i = 1 \qquad \text{(Eqn. 3)}$$

An average value of S_i for each species is measured experimentally at each of the 96 vertical pixel locations. At each pixel location an iterative procedure is then used to solve the five equations for the five unknowns ($X_{1,2,3,4}$ and T).

Some improvements have been made recently to this basic Raman measurement and calculation procedure. During daily calibration tests to determine K_i the laser power and gas temperature are measured and normalized to eliminate experimental variations. In addition, laser power is measured during all hot-fire tests where Raman signals are collected to eliminate similar variations. A correction has been applied to the bandwidth factor to account for the transmission characteristics of the individual bandpass filters. Finally, the number of individual data frames used to calculate an average Raman signal for each species has been increased from approximately 40 to 100 to reduce the amount of statistical variation in these measurements.

CURRENT ACTIVITIES

Test Series at Pc = 200 psia

Recent reports [2,3] have documented the results obtained over the past two years with a single thruster, emphasizing results with a chamber pressure of 500 psia. Because program plans include testing a twin thruster configuration at 200 psia (see the next section), the decision was made to conduct a test series with the single thruster operating at 200 psia. The objectives of this series are: (1) to develop a single thruster data set for comparison with the twin thruster data; (2) to compare the 200 psia results for both nozzle area ratios (AR = 6.0 and 3.3); (3) to compare results between the 5-inch and the 2-inch duct throat; and (4) to compare these experimental results with the CFD results being generated at NASA/Marshall.

One RBCC operating point was selected to perform these comparisons. This operating point, previously identified as Case 3 [2,3], uses the direct connect air supply configuration with the thruster operating at a mixture ratio of 8.0 and a chamber pressure of 200 psia. The nominal flowrates for this condition are 0.243 lb_m/sec of oxygen, 0.0304 lb_m/sec of hydrogen (rocket), 0.630 lb_m/sec of air, and 0.0184 lb_m/sec of hydrogen (afterburner).

Test Results: AR= 3.3 Thruster Nozzle Data

Figure 3 depicts Case 3 results with the AR= 3.3 nozzle at two window locations, 2.3 and 9.3 inches downstream of the thruster exit plane. The temperature plot (Figure 3a) has a typical jet mixing profile at the first window (x = 2.3 inches). The hot rocket exhaust (~ 2500-3000K) is prominent near the centerline, and the temperature asymptotically approaches ambient conditions (~ 300-400K) at the extremities. Note that the tails of these curves are not at the duct walls (+/- 2.5 inches), but rather near the edges of the original access windows (+/- 1.4 inches). Also note that the profile near the centerline is not well behaved for the x = 2.3 inch curve. This effect is due to the fact that at the first window location a significant amount of background light from the rocket plume is present. This background signal has been subtracted out, but the results are still affected by the background to some degree. The x = 9.3 inch temperature curve shows a significantly different characteristic. In fact, within the level of data scatter in the plot, it appears that the temperature is fairly uniform across the flowfield. The horizontal line in the plot is the calculated equilibrium temperature [7] of the air and the thruster propellants. This equilibrium temperature (2553K) compares well with the average value of the temperature data (2623K).

Similar observations can be made about Figure 3b, which includes the mole fraction profiles of nitrogen and water. Again the near-centerline values for the x= 2.3 inch curves are affected by the strong background signal, but the general shape of these curves indicate a well-defined rocket exhaust plume in the center with nearly pure air at the edges. At x= 9.3 inches the mole fraction curves indicate very uniform conditions, although the average values are not quite at the equilibrium conditions indicated by the horizontal lines.

Test Results: AR= 6.0 vs. 3.3 Thruster Nozzle Comparison

Figure 4 shows a comparison of the mole fraction data for the two thruster nozzle configurations at the x = 9.3 inches window. The AR= 6.0 data was generated in a 1998, prior to some of the current improvements in the data collection and analysis process. It is apparent from Figure 4 that the new procedure of collecting 100 frames of data results in significantly less pixel-to-pixel variation in the profile. More importantly, Figure 4 indicates that there is a significant difference between the mole fraction profiles at this location. While the AR= 3.3 data is essentially flat, there is a significant variation in the nitrogen and water profiles for the AR= 6.0 nozzle. Although the scatter in the individual data points somewhat obscures these trends, a second order polynominal curve fit has been added to the AR= 6.0 data to emphasize the trends. The trends are consistent with a jet mixing profile. The water mole fraction is largest near the centerline and it tapers off near the edges. The opposite trend is seen in the nitrogen profile.

Data is available for AR= 6.0 at other downstream window locations. Some of this data is plotted in Figure 5, along with the AR= 3.3 data at x = 9.3 inches. For the sake of clarity, only the nitrogen data is presented. As seen in Figure 5a, the nitrogen profile at x= 16.3 inches has flattened out somewhat. However, it is not until the x= 23.3 inch window (Figure 5b) that the mole fraction profile for AR= 6.0 resembles that of AR= 3.3 at x= 9.3 inches.

The implication of this data is that the primary and secondary streams mix much more rapidly for the AR= 3.3 configuration than for AR= 6.0. The only physical difference between these two cases is in the geometry of the nozzles at their exit plane. The rocket blockage height in both cases is 1.75 inches, but the nozzles are only 0.33 and 0.6 inches tall, respectively, at the exit.

The remaining distance (1.42 and 1.15 inches, respectively) acts as a rearward facing step, which will tend to set up a recirculation zone at the exit plane. The difference in base area between these two nozzle configurations (~ 25%) does not appear to be large enough to explain the significant change in mixing characteristics. Detailed CFD calculations may provide some insight into the differences between these two cases [8].

PLANNED ACTIVITIES

Twin Rocket Test Program

A practical RBCC propulsion system will probably incorporate multiple rocket ejectors in a duct. A correlation developed by Marquardt [1] for the optimum primary/secondary mixing length (L) is based on a number of flow and geometric parameters, including the number of thrusters (N) in the flowpath. The relationship between the mixing length and the number of thrusters is:

$$L \propto N^{-0.35} \qquad \text{(Eqn 4)}$$

Increasing the number of thrusters can significantly reduce the required duct mixing length. This reduction in mixing length will have a positive impact on the overall size and weight of a RBCC system.

In order to study the mixing effects of multiple ejectors, hardware has been designed and fabricated for the next test series. This hardware, as pictured in Figure 6, integrates two GH_2/GO_2 thrusters into the current duct configuration. Because of stress and cooling issues associated with these smaller thrusters, they will only be operated up to chamber pressures of 200 psia. The length of the twin thrusters will be identical to that of the single thruster. The height dimensions will be exactly half those of the single rocket. For instance, the nozzle throat height of the twin thrusters will be 0.050 inches. The combined duct blockage area and rocket propellant flow rates of the two thrusters will be equal to those of the current single thruster. Thus a direct comparison of results can be made between the twin rocket and the single rocket configurations.

In addition, the hardware has been designed to allow three different thruster spacings. The thruster centerline to centerline distance (b in Figure 6) can be set at 1.75, 2.50 or 3.25 inches. In all cases the thrusters are symmetric about the centerline of the 5 inch high duct.

These three thruster spacing options can be used to study geometric effects on the mixing process. The different spacing configurations should provide enough variation to study the effect of plume-to-plume interaction on the mixing process. This effect plus any others observed should provide some insight into the factors that affect optimum thruster spacing. The ultimate goal is to couple the physical insight gained from these measurements into a mixing model that can be used in more complex CFD models or global performance prediction models.

SUMMARY

Continued testing of the single GH_2/GO_2 thruster configuration of the RBCC test article has led to improvements in the Raman data collection and analysis process. These improvements will be used in the next phase of testing, the twin thruster test series.

Results from the 200 psia test series indicate a very rapid primary/secondary mixing process in the duct when the AR= 3.3 nozzle is installed on the thruster. The significant difference between the mixing data for this nozzle and AR= 6.0 nozzle is not well understood. Additional insight may be gained through a combination of additional testing and CFD analysis.

ACKNOWLEDGEMENTS

The authors acknowledge funding from NASA/Marshall Space Flight Center under NASA Contract NAS8-40890. The authors thank Joe Ruf and Jeff West of NASA/Marshall who are performing CFD calculations that are complementary to this experimental work. Finally the authors thank Larry Schaaf and Mayumi Greene for their assistance in conducting these experiments.

REFERENCES

[1] Odegaard, E. A. and Stroup, K. E., "1966 Advanced Ramjet Concepts Program, Volume VIII- Ejector Ramjet Engine Tests- Phase I," The Marquardt Corporation, Technical Report AFAPL-TR-67-118 Volume VIII, January 1968.

[2] Lehman, M., Pal, S. and Santoro, R. J., "RBCC Project Summary: Experimental Investigation of Ejector Mode," Propulsion Engineering Research Center 11th Annual Symposium on Propulsion, University Park, November 18-19, 1999.

[3] Lehman, M., Pal, S. and Santoro, R. J., "Experimental Investigation of the RBCC Rocket-Ejector Mode," AIAA Paper 2000-3725, 36th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Huntsville, Alabama, July 17-19, 2000.

[4] Eckbreth, A. C., Laser Diagnostics for Combustion Temperature and Species, Energy and Engineering Sciences Series, eds. A. K. Gupta and D. G. Lilley, Vol. 7, Abacus Press, Cambridge, Massachusetts, 1988.

[5] Foust, M. J., Pal, S., and Santoro, R. J., "Gaseous Propellant Rocket Studies Using Raman Spectroscopy," AIAA Paper 96-2766, 32nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Lake Buena Vista, Florida, July 1-3, 1996.

[6] Yeralan, S., Pal, S., and Santoro, R. J., "Major Species and Temperature Profiles of LOX/GH2 Combustion," AIAA Paper 97-2940, 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Seattle, Washington, July 6-9, 1997.

[7] Gordon, S. and McBride, B., Computer Program for Calculation of Complex Chemical Equilibrium Compositions, Rocket Performance, Incident and Reflected Shocks, and Chapman-Jouguet Detonations, NASA SP-273, NASA Scientific and Technical Information Office, Washington, D.C., 1971.

[8] West, J. and Ruf, J., "FDNS CFD Code Benchmark for RBCC Ejector Mode Operation: Continuing Toward Dual Rocket Effects," Propulsion Engineering Research Center 12th Annual Symposium on Propulsion, Cleveland, Ohio, October 26-27, 2000.

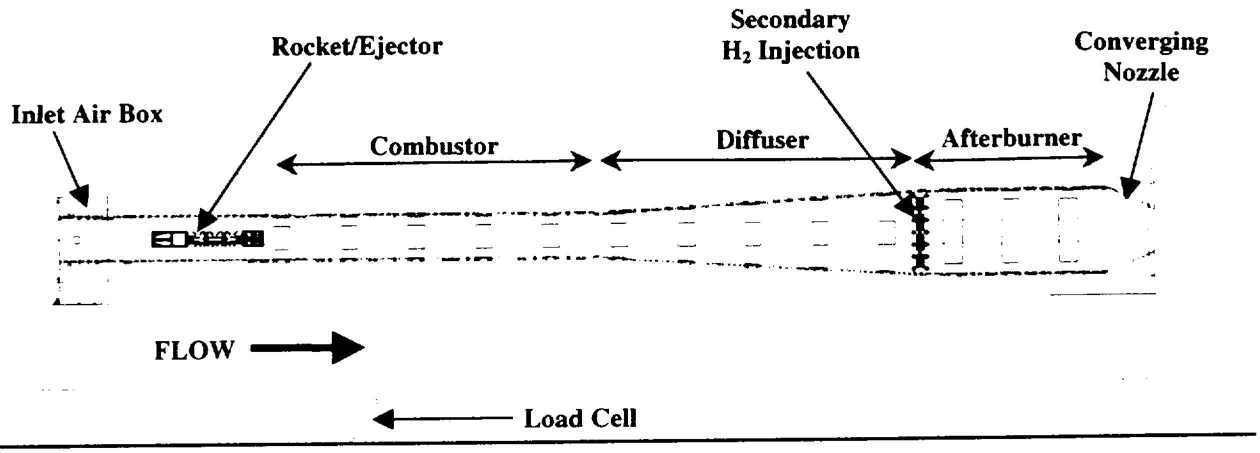

Figure 1- RBCC Test Article, Direct Connect Configuration (single thruster, 5-inch throat).

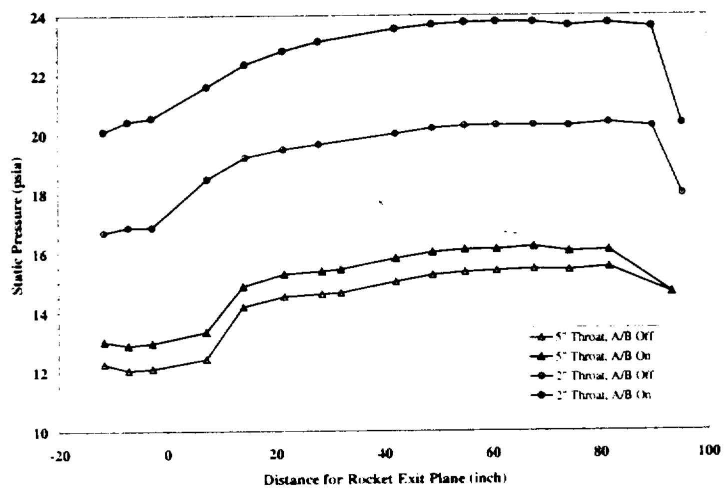

Figure 2- Static Pressure Profile in Duct for 2" Throat and 5" Throat (with and without afterburner [A/B])

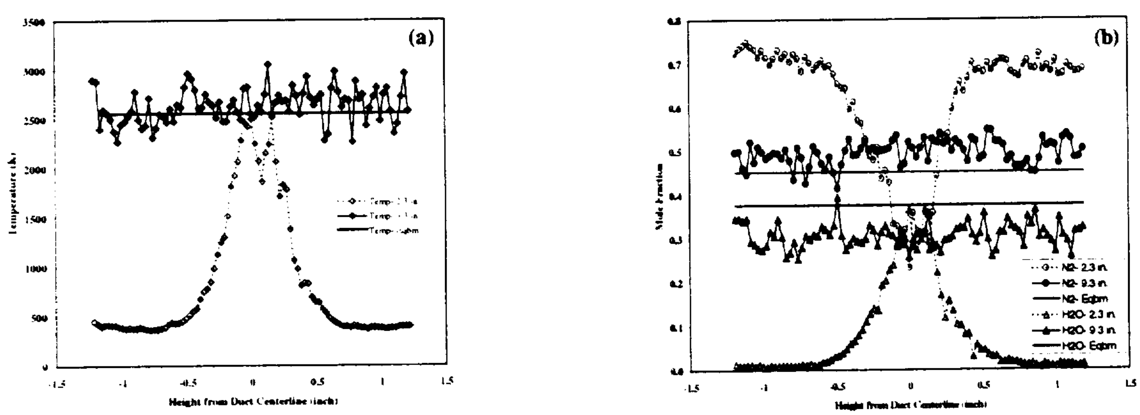

Figure 3- Case 3 Raman Measurements with AR= 3.3 Nozzle
(a) Temperature Profile and (b) Nitrogen and Water Mole Fraction Profiles

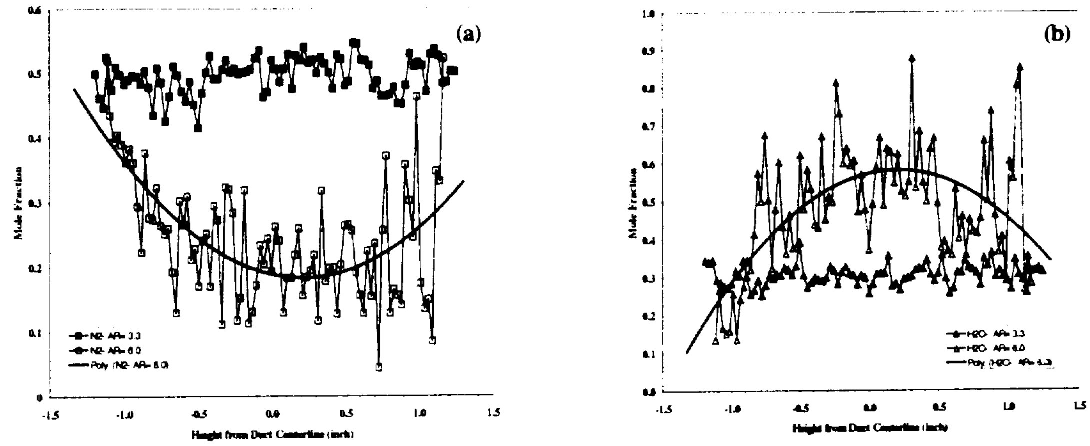

Figure 4- Mole Fraction Comparisons Between AR= 3.3 and AR= 6.0 Nozzles at X= 9.3 inches
(a) Nitrogen Profile and (b) Water Profile

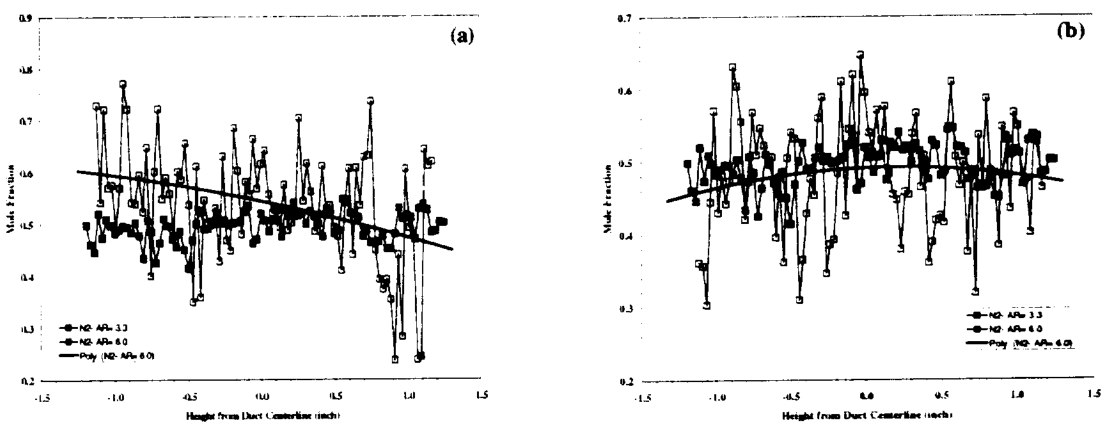

Figure 5- Nitrogen Mole Fraction Comparisons Between AR= 3.3 Nozzle (X= 9.3 in.) and AR= 6.0 Nozzle
(a) X= 16.3 inches and (b) X= 23.3 inches

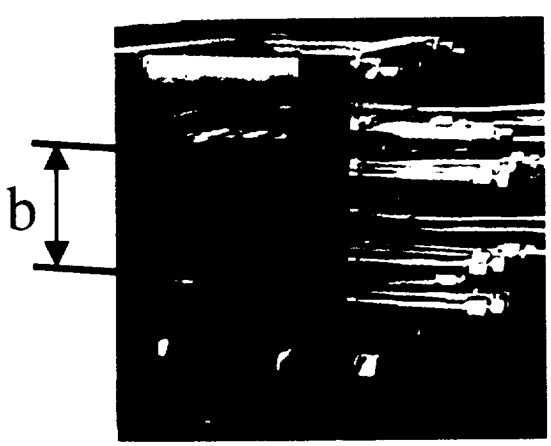

Figure 6- Aft view of twin rocket configuration.